This document is guaranteed to be current only to issue date.

Some Mars Global Surveyor documents that relate to flight operations are under revision to accommodate the recently modified mission plan.

Documents that describe the attributes of the MGS spacecraft are generally up-to-date.

542-200

Mars Global Surveyor

Spacecraft Requirements

REVISION A

September 10, 1996

Jet Propulsion Laboratory
California Institute of Technology

JPL D-11509

542-200

Mars Global Surveyor

Spacecraft Requirements

REVISION A

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September 10, 1996

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JPL D-11509

Revision Tracking Log

Exhibit Revision Letter	Revision Date	Revision Description	Effected Pages
A	11/15/94	Change to spacecraft injected and structural mass requirements. Incorporates ECR 89611.	3-1
A	4/7/95	Change of total mission delta-V required. Incorporates ECR 89620.	3-2
A	10/6/95	Change of opening launch period and addition of support for two launch windows per day. Incorporates ECR 89626.	3-1
A	7/1/96	Change in Relay Operations Phase; Reduction of on-orbit mission life from 5 to 2.5 years. Incorporates ECR 89638.	1-2, 3-1
A	7/1/96	Change Launch Period to November 6-25, 1996. Incorporates ECR 89638.	3-1
A	7/1/96	Reference to payload ICDs into section 4. Revision of temp and power data per ICDs. Incorporates ECR 89638.	2-2, 4-1, 4-2, 4-4, 4-5
A	7/1/96	Clarification of Radio Science data collection requirements. Incorporates ECR 89638.	4-2
A	9/10/96	Restores original 5-year on-orbit mission life. Reflects ECR 89638 rev. A	1-2, 3-1

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SECTION 1

INTRODUCTION

1.1 SCOPE

This document establishes the requirements for the Mars Global Surveyor spacecraft. Section 1 contains definitions; all other sections contain requirements.

1.2 TERMS AND ABBREVIATIONS

The following definitions of terms and abbreviations are used in this document.

1.2.1 Terms

(1) Spacecraft Bus: The spacecraft bus comprises all assemblies, engineering subsystems, associated flight software, and miscellaneous hardware that constitute the spacecraft without the science instruments. It includes all propellant required by the spacecraft after injection, as well as government-furnished property (GFP) used, excluding the payload.

Included as a part of the spacecraft bus is a stub adapter or other means which provide the spacecraft side of the interface with the L/V-to-S/C separation plane.

(2) Payload: The payload consists of the complement of the science instruments provided as GFP.

(3) Spacecraft: The spacecraft is the composite of the spacecraft bus and the payload after integration.

(4) Delta Launch Vehicle: The Delta launch vehicle is the Delta II 7925 configuration consisting of a solid motor augmented first stage, a liquid propulsion restartable second stage, a spin-stabilized solid propulsion third stage, and a payload fairing.

(5) Delta Third Stage: The third stage consists of all assemblies, subsystems, associated flight software, and miscellaneous hardware used to inject the spacecraft onto the heliocentric trajectory to Mars from low-Earth orbit.

(6) Ground Support Equipment: The ground support equipment (GSE) comprises the ground hardware and software required to verify and check out the elements of the spacecraft during fabrication, assembly, testing, prelaunch, launch, and post-launch operations.

(7) Mission Operations System: The mission operations system (MOS) consists of the complement of ground hardware, software, procedures, and personnel used to conduct flight operations.

(8) Launch Phase: The launch phase extends from the start of the launch countdown to separation of the spacecraft from the Delta third stage.

(9) Cruise Phase: The cruise phase extends from separation of the spacecraft from the third stage to the initiation of the orbit insertion sequence. It includes

initial checkout of the spacecraft, and the required trajectory correction maneuvers (TCM) and calibrations.

(10) Orbit Insertion Phase: The orbit insertion phase extends from the initiation of the Mars orbit insertion (MOI) sequence until the spacecraft is declared ready to collect mapping phase science data. During this phase, the insertion into the mapping orbit will be accomplished, and the spacecraft configured to commence science data collection.

(11) Mapping Phase: The mapping phase is the time period during which science data are returned from the Mars mapping orbit. It extends 687 Earth days after the end of the orbit insertion phase, from approximately January 1998 through November 1999.

(12) Relay Operations Phase: Upon completion of mapping phase, the spacecraft shall function primarily as a relay satellite for the remainder of the five Earth-year on-orbit design life. This phase is from approximately November 1999 through November 2002.

(14) DSN Pass: A DSN pass is defined as a single continuous 10-hour period during which a Deep Space Network (DSN) station is available for spacecraft tracking, commanding, and telemetry return.

(15) Quarantine Orbit: Near-circular, 400-km index altitude orbit.

1.2.2 Abbreviations

AGC	automatic gain control
AU	astronomical unit
b/s	bits per second
CCAFS	Cape Canaveral Air Force Station
CCSDS	Consultative Committee for Space Data Systems
C&DH	command and data handling
CDU	command detector unit
dBc	decibel referred to carrier level
dB/K	decibel per Kelvin
dBm	decibel referred to 1 milliwatt
deg	degree
DSN	Deep Space Network
DSS	Deep Space Station
EIRP	effective isotropic radiated power
ELS	eastern launch site
EMC	electromagnetic compatibility
EMI	electromagnetic interference
ENG	engineering-only telemetry data stream
ER	electron reflectometer
ERR	eastern range regulation
ESMC	Eastern Space and Missile Center
ET	ephemeris time
FED-STD	federal standard
FOV	field of view
GFP	government-furnished property
GSE	ground support equipment
G/T	gain/temperature

Hz	hertz
IAU	International Astronomical Unit
ICD	interface control document
ID	identification
IFOV	instrument field of view
JPL	Jet Propulsion Laboratory
KABLE	Ka-band link experiment
kb/S	kilobits per second
keV	thousand electron volts
kHz	kilohertz
KSC	Kennedy Space Center
ks/s	kilosymbols per second
L	launch
m	meter
MAG	magnetometer
MEF	maximum expected flight
MDA	McDonnell Douglas Aerospace
MIL-STD	military standard
MGS	Mars Global Surveyor
MOC	Mars orbiter camera
MOI	Mars orbit insertion
MOLA	Mars orbiter laser altimeter
MOS	mission operations system
MR	Mars relay
mrad	milliradian
mW	milliwatt
N/A	not applicable
NASA	National Aeronautics and Space Administration
NHB	NASA Handbook
nT	nanotesla
OTM	orbit trim maneuver
PDS	payload data subsystem
POR	power-on reset
RF	radio frequency
RFP	request for proposal
RS	radio science
RSS	root sum square
RTC	real-time commands
SBRC	Santa Barbara Research Center
SDS	spacecraft data storage
S/C	spacecraft
S&E-1	science and engineering data stream 1
S&E-2	science and engineering data stream 2
SEE	single-event effect
SEL	single-event latchup
SEU	single-event upset
SFOC	Space Flight Operations Center
SPE	static phase error
SRFOV	stray radiation field of view
SSC	stored sequence commands
TCM	trajectory correction maneuver
TES	thermal emission spectrometer
USO	ultrastable oscillator
VDIR	view direction

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SECTION 2

APPLICABLE DOCUMENTS

The following documents apply to the extent specified in this document, with revision number/date of issue as specified in Exhibit I.

2.1 LAUNCH SYSTEM

Document No.	Title
ERR 127-1	Eastern Range Regulation - Range Safety
(NO NUMBER)	DEPARTMENT OF THE AIR FORCE, 45th SPACEWING (AFSPACECOM) letter dated 23 November 1993: Subject: Interim Safety Requirements for Design, Test, and Ground Processing of Flight Graphite Epoxy (GR/EP) composite overwrapped Pressure Vessels (COPVs) at the Kennedy Space Center (KSC), Cape Canaveral Air Force Station (CCAFS), and Vandenberg Air Force Base (VAFB).
GP-1098	KSC Ground Operations Safety Plan
K-CM-05.3	Guide for Payload Processing at KSC
MDC H3224	Delta II Payload Planners Guide

2.2 PRODUCT ASSURANCE

Document No.	Title
FED-STD 209	Clean Room and Work Station Requirements, Controlled Environments
JPL D-11510	Performance Assurance Provisions
JPL D-11513	Spacecraft Environmental Estimates

2.3 SPACECRAFT

Document No.	Title
CCSDS	Command and Telemetry Standards Published by the Consultative Committee for Space Data Systems
DM514438	Deep Space Command Detector Unit National Aeronautics and Space Administration Design Requirement For
JPL 810-5	Deep Space Network/Flight Projects Interface Design Handbook
JPL D-1678	Payload Data Subsystem Characteristics
JPL D-3419, vol.1	PDS Functional Requirements Document

JPL D-6145	Spacecraft Bus/Command Detector Unit (CDU): Interface Control Document
SE-003	Spacecraft Bus / Mars Relay Interface Control Document
SE-004	Spacecraft Bus / Mars Orbiter Laser Altimeter Interface Control Document
SE-005	Spacecraft Bus /Mars Orbiter Camera Interface Control Document
SE-006	Spacecraft Bus /Thermal Emission Spectrometer Interface Control Document
SE-007	Spacecraft Bus / Magnetometer and Electron Reflectometer Interface Control Document
SE-008	Spacecraft Bus / Ultrastable Oscillator Interface Control Document
JPL D-6623	Spacecraft Bus/Unique Interface Control Document for the Payload Data Subsystem
JPL D-11513	Spacecraft Environmental Estimates
JPL D-11514	Trajectory Characteristics Document
MIL-STD-1576	Electroexplosive Subsystem Safety Requirements and Test Methods for Space Subsystems

SECTION 3

SYSTEM REQUIREMENTS

3.1 MISSION LIFETIME AND RELIABILITY

3.1.1 Lifetime

The spacecraft shall have a design lifetime of at least 5 years after the establishment of the mapping orbit and be capable of supporting science data collection in the mapping phase, supporting Mars relay operations during the relay operations phase, and achieving a quarantine orbit if necessary.

3.1.2 Reliability

Appropriate block, functional, or alternative mode redundancy shall be employed to avoid single-point mission-critical failures. Specific exceptions to this requirement shall be identified and evaluated; they will be approved only if the failure mechanism is found to be acceptably improbable.

A mission-critical failure is defined to be a failure that results in the permanent loss of data from more than one scientific instrument during the mapping phase, loss of the relay capability during the relay phase, the failure to achieve and maintain the proper orbit or pointing control to within specified tolerances, the loss of science-critical engineering telemetry required for attitude determination, or the failure to achieve the quarantine orbit (if required) prior to the end of the mission.

The design shall also accommodate mission operation in degraded modes. A degraded mode of operation is defined to be one in which the primary scientific objectives of the mission can still be met, but at the expense of a loss of some scientific data and/or an increase in the complexity of the mission operations.

3.2 LAUNCH

The spacecraft will be carried into Earth orbit by the Delta II 7925 (Delta II) and injected into the trans-Mars trajectory by the Delta third stage. The total injected spacecraft mass, plus the despin hardware mass, with required adjustments for the payload attach fitting, shall not exceed 1064 kg. Launch shall occur during the Mars opportunity of November 6-25, 1996, with launch possible on any day of the launch period. During the period of November 6, 1996 through November 15, 1996, the spacecraft shall be capable of supporting two instantaneous daily launch windows which will have a minimum separation of 48 minutes and a maximum separation of 73 minutes.

3.3 NAVIGATION

3.3.1 Cruise Phase Maneuvers

Trajectory correction maneuvers (TCMs) will be executed between injection and arrival at Mars to correct injection errors and to adjust the arrival conditions.

The spacecraft shall be capable of executing up to four TCMs at any time during the cruise phase. The spacecraft shall be capable of executing the velocity increment for a TCM in any inertial direction. TCMs will not be planned earlier than injection plus 15 (I+15) nor later than 20 days prior to Mars Orbit Insertion (MOI-20.)

3.3.2 Mapping Orbit Initiation

The spacecraft shall be delivered from a northern approach trajectory to Mars into capture orbit with a propulsive maneuver. The spacecraft shall then transition to a 2:00 pm local mean solar time sun-synchronous mapping orbit with propulsive maneuvers and aerobraking.

The spacecraft shall be capable of performing periapsis altitude adjustment maneuvers as frequently as once a day during the aerobraking phase. The spacecraft shall be capable of establishing and operating within specification in any mapping orbit within the range of orbital elements shown in table 3-1.

The spacecraft shall be capable of providing a total mission delta-V of 1290 m/s, inclusive of finite burn losses from thrust vector misalignments, gravity losses, and all other maneuver inefficiencies.

After mapping orbit establishment but prior to commencement of mapping operations, a 7 day period will be allocated for data acquisition for Mars gravity calibrations. Spacecraft design shall be such that the spacecraft is as free from non-gravitational accelerations as possible during this gravity calibration period.

Table 3-1. Mapping Orbit Mean Elements[a]

Element	Minimum Semimajor Axis (377.9 km index altitude)[b]	Maximum Semimajor Axis (400 km index altitude)
Semimajor Axis	3775.1 km	3797.2 km
Eccentricity	0.0072 ± 0.007	0.0072 ± 0.007
Inclination	92.87 deg	92.93 deg
Ascending Node[c]	$-39.1664 + 0.524\,\Delta T \pm 3$ deg[d]	
Argument of Periapsis	-90 ± 10 deg	
Coordinate System	Mars mean equator and IAU vector of epoch	
Epoch	ΔT Earth days past 1/1/1998 at 0000 hours ET	

[a] Mean orbital elements consistent for truncated Balmino gravity field (unnormalized.)

[b] Index altitude is measured with respect to Mars equatorial radius of 3397.2 km.

[c] Ascending node located on the night side of Mars.

[d] Sun-synchronous orbit requirement only applies to mapping phase.

3.3.3 Orbit Maintenance

The spacecraft shall be capable of performing orbit trim maneuvers (OTMs) as frequently as every 7 days. The total OTM ΔV requirements shall be taken from Figure 3-1. The spacecraft shall be capable of executing an OTM velocity increment in any inertial direction.

The relay orbit is not required to be sun-synchronous.

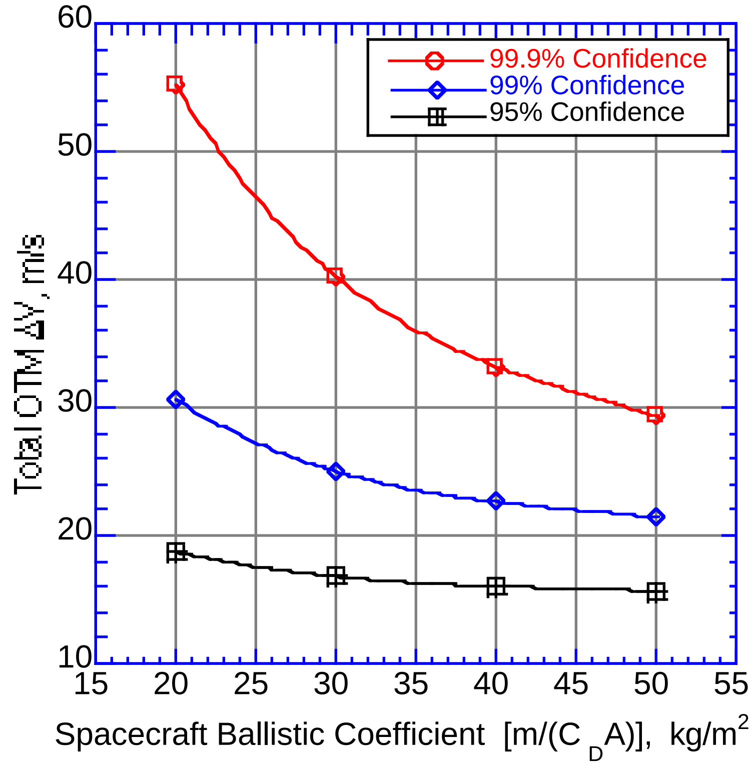

Figure 3-1. Total OTM ΔV Requirement (Mapping and Relay Phases)

3.3.4 Maneuver Execution Errors.

The spacecraft shall be capable of controlling maneuver magnitude and direction per the limits set forth in table 3-2. For each component of each maneuver, the allowable error shall be less than the RSS of the 3σ fixed and proportional error limits shown.

Table 3-2. Maneuver Execution Error Limits (3σ)[a]

Error Source	Error Limit
Proportional Magnitude	2.0%
Fixed Magnitude	0.05 m/s
Proportional Side Velocity (Total)[b]	2.5%
Fixed Side Velocity (Total)	0.01 m/s

[a] Maneuver execution errors have two components: (1) magnitude errors that are measured parallel to the desired impulse direction and (2) side velocity errors that are measured perpendicular to that direction. The error model assumes that the error in each component is made up of a fixed error, which is independent of the impulse magnitude, and a proportional error, which is proportional to the impulse magnitude.

[b] The proportional side velocity error corresponds to a pointing error of 25 mrad.

3.3.5 Spacecraft Ephemeris in the Mapping Orbit

3.3.5.1 Ephemeris Prediction. During the mapping phase, if any spacecraft function depends on the ability of the MOS to predict the ephemeris of the spacecraft, the analysis of that function shall be based on the ephemeris prediction errors given in Table 3-3.

Table 3-3. Mapping Orbit Ephemeris Prediction Errors[a]
Position

Position Component	3-σ Uncertainty in the Predicted Position of the Spacecraft after 14 Days (km)[b]
Downtrack	25
Crosstrack	9
Radial	8

[a] Excludes solar conjunction

[b] Ephemeris prediction errors during the period surrounding Mars perihelion will increase due to atmospheric density errors. During these periods more frequent emphemeris updates will be required to achieve the values in this table.

3.3.5.2 Ephemeris Reconstruction. During the mapping phase, if any spacecraft capability depends on the ability of the MOS to provide after-the-fact knowledge of the ephemeris of the spacecraft, the analysis of that capability shall be based on the ephemeris reconstruction errors given in Table 3-4.

Table 3-4. Mapping Orbit Ephemeris Reconstruction Errors[a]

Position Component	3-σ Uncertainty in the Reconstructed Position of the Spacecraft (km)
Downtrack	9
Crosstrack	5
Radial	2

[a] Excludes solar conjunction

3.3.6 Propellant Load

The flight propellant load, including pressurant, shall be sufficient to accommodate propulsion system performance uncertainties and provide the specified ΔV with a 99% confidence.

3.3.7 Maneuver Reconstruction Telemetry. The spacecraft shall provide engineering telemetry to allow reconstruction of maneuver, attitude control and momentum management events for navigational and radio science purposes.

3.3.8 Tracking Data

The spacecraft shall be capable of supporting the navigational tracking schedule shown in Table 3-5, excluding periods of maneuvers and solar conjunction.

Table 3-5. Navigation Tracking Requirements

Mission Phase	**Daily Tracking Coverage**	**Data Type**
Cruise		
Initial Acquisition completion to L+30 days	24 hours	2-way coherent Doppler & ranging
L+30 to MOI-90 days	10 hours[a]	2-way coherent Doppler & ranging
TCM-3 to TCM+3 days	24 hours	2-way coherent Doppler & ranging
MOI-90 to MOI-30 days	20 hours	2-way coherent Doppler & ranging
MOI-30 to MOI	24 hours	2-way coherent Doppler & ranging
Orbit Insertion[b]		
MOI to mapping orbit	24 hours	2-way coherent Doppler & ranging
Mapping		
1. Daily	One DSN pass	2-way coherent Doppler & ranging
2. Every 3rd day	Two DSN passes[c]	2-way coherent Doppler & ranging
3. A period of 4 weeks duration, nominally centered on the date of the edge-on orbital configuration.	Two DSN passes	2-way coherent Doppler & ranging
Relay Phase		
Every second day	One DSN pass	2-way coherent Doppler & ranging

[a] A single daily DSN pass and 2 consecutive DSN passes per week.

[b] Within limits of power available.

[c] Every third day the 2nd pass will be reserved for real-time science and engineering data transmission.

3.3.8.1 Antenna Phase Center Motion. Mechanically induced antenna phase center motion shall not be greater than 0.1 mm/sec (1σ) with respect to the spacecraft center of gravity over a sixty second integration time, except during momentum management periods.

3.3.9 Planetary Protection & Quarantine Orbit

If any part of the spacecraft is jettisoned after injection into the trans-Mars trajectory, it shall be shown by analysis that the probability of accidental impact of any part of the spacecraft on Mars shall be less than 10^{-2} up to 20 years after launch and 0.05 for an additional 30 years

At the end of the mission, the spacecraft shall be capable of being raised to the quarantine orbit.

3.4 OPERABILITY

3.4.1 Commands

The spacecraft shall be capable of performing its required functions through the use of on-board stored operating programs, stored sequence commands and ground-transmitted real-time commands.

3.4.1.1 Prelaunch & Launch Phase. During prelaunch and after the spacecraft and fairing are mated to the launch vehicle, the spacecraft shall be capable of receiving mission-critical commands through the spacecraft umbilical. The spacecraft design shall provide for operation for at least 48 hours following separation without the need for ground commands.

3.4.1.2 Emergency Commands. In case of a failure that disrupts normal ground-to-spacecraft communications, the spacecraft shall provide for autonomous onboard action that makes it continuously receptive to low-rate commands

3.4.1.3 Time Criticality. The spacecraft shall not require time-critical transmission of ground commands.

3.4.2 Data Return

3.4.2.1 Launch Phase. Any spacecraft quantity or function capable of being updated by the ground during the launch phase shall be verifiable in telemetry. The spacecraft shall be capable of storing launch and ascent spacecraft telemetry and subsequently downlinking the stored telemetry stream.

3.4.2.2 Cruise and Orbit Insertion Phases. During these phases, the spacecraft shall be capable of gathering and returning data to characterize the performance and health of the spacecraft. These data may be returned either in real-time or non-real time.

The spacecraft shall be capable of gathering and returning data to support in-cruise payload calibration and science data return. These data can be returned either in a real-time or a non-real-time mode.

3.4.2.3 Mapping Phase. During the mapping phase of the mission, the spacecraft shall be capable of continuously gathering, storing and transmitting science and engineering data.

The spacecraft shall be capable of transmitting these data to Earth during a daily DSN pass which will allow a total of 4.5 noncontinuous hours of downlink. In addition, the spacecraft shall be capable of returning real-time Radio Science data (non-coherent carrier using the USO without telemetry modulation) for up to ten minutes per orbit during scheduled DSN passes.

3.4.2.4 Relay Phase. During the relay phase of the mission, the spacecraft shall be capable of continuously gathering, storing and transmitting relay data as described in Section 4.

3.4.3 Engineering Data Requirements

3.4.3.1 Engineering Telemetry. The spacecraft shall provide for a minimum of two telemetry subcommutation maps which shall be selectable and modifiable by ground command.

3.4.3.2 Science Data Characterization. The spacecraft shall provide engineering data for incorporation in the telemetry stream that enables correlation of the spacecraft's state, performance, attitude and environment with the science data.

3.4.3.3 Health Assessment. The spacecraft shall provide data to perform an assessment of the current health of the spacecraft through the telemetry downlink during each DSN pass.

3.4.3.4 Propellant Status. The spacecraft engineering telemetry shall provide measurements to allow determination of the quantity of propellants remaining on the spacecraft.

3.4.3.5 Memory Contents and Validation. The spacecraft shall be capable, via ground command, of reading out any or all of its memory through telemetry. The spacecraft shall also provide for validating its memory without having to perform a complete memory readout.

3.4.3.6 Command Accountability. The spacecraft shall provide information in the telemetry stream to allow confirmation of command receipt and execution.

3.4.3.7 Emergency Telemetry. In the event of abnormal conditions onboard the spacecraft, the emergency telemetry data volume within a single DSN pass shall be sufficient to determine the spacecraft state.

The emergency telemetry data transfer frame length shall be of sufficient size to allow a minimum of 5 full frames to be received by the DSN during a single orbit.

3.4.4 Autonomous Operations

Due to the adverse effect of interplanetary distances and solar conjunction on the commandability of the spacecraft, autonomous spacecraft operations shall be provided for as follows:

3.4.4.1 Normal Autonomous Operations. The spacecraft shall provide for autonomous management of the following:

(1) orbital position determination with respect to eclipse. While in orbit around Mars, the spacecraft shall have the capability to autonomously detect eclipse entry. Upon eclipse entry, the spacecraft shall initiate execution of a stored command sequence designed for eclipse ingress, and upon eclipse egress shall initiate execution of an independent stored command sequence designed for eclipse egress. It shall be possible to enable/disable this capability by real-time or stored sequence command.

(2) battery state-of-charge.

(3) nadir pointing during mapping and relay phases.

(4) spacecraft momentum.

(5) thermal control of subsystems and payload where small thermal time constant would require real-time corrective action in less than 72 hours.

(6) high-gain antenna pointing by stored program.

(7) data storage.

(8) payload sun-avoidance during spacecraft slews for propulsive maneuvers or slews to sun acquisition or coning orientations.

3.4.4.2 Autonomous Fault Protection.

(1) Spacecraft Health. In the event of a failure, the spacecraft shall be capable of autonomously maintaining for at least 72 hours the minimum functions required for safe system operation, including payload protection from abnormal spacecraft states or attitudes.

(2) Telecommunications. The spacecraft shall provide for autonomous initiation of continuous low-rate telemetry in the event of an on-board failure which would interfere with normal high-gain antenna communications. The spacecraft shall provide for autonomous initiation of emergency telemetry in the event that no commands are received within a ground-selectable time period.

(3) Power-On Reset (POR). The spacecraft shall enter a known and verifiable state upon spacecraft bus POR.

3.4.4.3 Operation During Solar Conjunction. The spacecraft shall be able to function without ground commands and maintain the minimum functions required for safe system operation during the period when the Earth-Spacecraft-Sun angle is less than 2 degrees.

3.5 ENVIRONMENTAL REQUIREMENTS

The spacecraft shall be designed to meet the functional requirements as specified in this document when operating in the expected mission environment described in JPL D-11513 "Spacecraft Environmental Estimates", with the design margins specified herein, and when under test in accordance with the provisions of JPL D-11510, "Performance Assurance Provisions". Except where specified otherwise, all environmental design requirements shall equal or exceed the corresponding protoflight test requirements in JPL D-11510. Table 3-6 summarizes the required design margins to be applied to the environmental estimates of JPL D-11513.

3.5.1 Ground Operations and Handling

3.5.1.1 Particulate Contamination. The spacecraft shall be assembled and maintained in a class 100,000 clean room per Federal Standard 209.

3.5.1.2 Thermal/Humidity Environment. The spacecraft shall be designed to operate within specifications after exposure to the ambient thermal and humidity environment expected during ground operations, including assembly, handling, transportation, and storage. If provisions are made to maintain the temperature and humidity of assemblies within that defined for the controlled environment as specified in JPL D-11513, then values for the controlled environment should be used for design, and design verification tests for the controlled environment are not required. If the temperature and humidity of assemblies are uncontrolled, then the values for the uncontrolled environment as specified in JPL D-11513 should be used for design, and design verification tests at the upper and lower levels of the uncontrolled environment are required.

3.5.1.3 Ground Operations and Handling Dynamics. The spacecraft shall be designed to operate within specifications after exposure to the vibration, acceleration, and shock experienced during ground operations and handling. Special shipping and handling equipment shall be utilized so that transportation levels do not exceed flight levels.

3.5.1.4 Electromagnetic Compatibility. The spacecraft shall be designed and fabricated to be compatible with the external and self-induced electromagnetic environments that will exist and to function without degradation when operated in any combination of modes. Each assembly shall be designed and constructed so as not to cause electromagnetic interference (EMI) to other assemblies or to itself. The design shall provide a susceptibility margin of 9 dB above expected electromagnetic radiated and conducted environments and shall provide an emission margin at least 9 dB (20 dB for pyro devices) below the expected environment specified in JPL D-11513.

The spacecraft shall be compatible with and shall not produce levels in excess of the Delta II electromagnetic interference (EMI) and electromagnetic compatibility (EMC) requirements contained in MDC H3224 as specified in JPL D-11513.

Table 3-6 Summary of Environmental Design Requirements

ENVIRONMENT	DESIGN REQUIREMENT
Dynamics	
Sine Vibration[a]	MEF[b] x 1.5, 5-100 Hz
Random Vibration[a]	MEF + 4dB[c]
Acoustics	MEF + 4dB
Pyro Shock[a]	MEF + 4dB
Thermal and Vacuum	
Thermal/Vacuum	Greater of -30/85°C or MEF ±35°C
Thermal Shock[d]	MEF dT/dt x 1.5
Launch Pressure Profile	MEF x 1.5
EMC/EMI	
Conducted Susceptibility	MEF + 9 dB
Conducted Emissions	MEF
Radiated Susceptibility	MEF + 9 dB[e]
Conducted Emissions	MEF

[a] Any axis
[b] MEF = Maximum expected flight levels from JPL D-11513
[c] For random vibration, MEF shall be at at least "minimum workmanship levels."
[d] Only for assemblies with dT/dt>50° C/Hour
[e] For pyro devices a design margin of 20 dB is required

3.5.1.5 Explosive Atmosphere. The spacecraft shall operate safely in the explosive atmosphere that may exist during launch preparations and launch.

3.5.2 Launch phase

3.5.2.1 Dynamics. The spacecraft shall function within specification after exposure to the acoustic, vibration, shock, and acceleration environments experienced during launch and separation of the spacecraft from the Delta II as specified in JPL D-11513 with a design margin of 1.5 for sine vibration and 4 dB for random vibration, acoustics and shock.

Those elements of the spacecraft that are required during the launch phase shall function within specification during exposure to the launch phase dynamic environment.

3.5.2.2 Thermal. The spacecraft shall maintain temperatures within the environmental estimates of JPL D-11513 when exposed to the environmental extremes encountered during these phases as specified in that document. The spacecraft assembly designs shall be based on the environmental extremes as specified in JPL D-11513 margined by ±35°C.

3.5.2.3 Launch Pressure Profile. The spacecraft shall withstand the pressure environments and pressure changes experienced during launch as specified in JPL D-11513 with a design margin of 1.5.

3.5.2.4 Neutral Atomic Oxygen Contamination. Surfaces exposed to the atmosphere in low Earth orbit shall be designed to perform within specification during and after exposure to neutral atomic oxygen.

3.5.2.5 Contamination. Contamination due to either the Delta II, the spacecraft, or the environment shall not unacceptably degrade the spacecraft performance at any time during the mission.

3.5.3 Cruise, Orbit Insertion, and Mapping Phases

3.5.3.1 Thermal. The spacecraft shall maintain temperatures within the environmental estimates of JPL D-11513 when exposed to the environmental extremes encountered during these phases as specified in that document. The spacecraft assembly designs shall be based the environmental extremes as specified in JPL D-11513 margined by ± 35°C.

3.5.3.2 Solid Particles. The spacecraft shall employ protective means or damage-tolerant design against failures due to solid particle (micrometeoroids) impact during the mission. For external cables, pressurized lines and internal components, protection shall provide a 0.95 probability of no penetration based on a Poisson distribution and the expected meteoroid fluence as specified in JPL D-11513. The expected meteoroid fluence shall be margined by a factor of three for surface material effects.

3.5.3.3 Magnetic Field. The spacecraft shall be designed to operate within specification when exposed to the ambient magnetic fields encountered during the mission as specified in JPL D-11513 .

3.5.3.4 Radiation and Single Event Effects. The spacecraft shall be designed to accommodate the natural particle radiation encountered during the mission as specified in JPL D-11513 . The levels for radiation design of assemblies to total ionizing doses shall be expected radiation values margined by a factor of two. For surface effects the design margin shall be 1. The spacecraft shall be designed so that single-event effects (SEEs), including single event upsets (SEUs) and single event latchups (SELs), do not cause mission-critical failures.

3.5.3.5 Electrostatic Charge and Discharge. The spacecraft design shall minimize the possibility of internal electrostatic discharge, and be immune to any electrostatic discharges occurring on external surfaces. Metallized layers of thermal blankets shall be strapped together and grounded to the spacecraft structure. All electrical cables from the interior of the spacecraft shall be shielded with the shield conductively grounded to the spacecraft structure.

3.5.3.6 Aerobraking Environment. the spacecraft shall be able to accomodate the aerodynamic heating rates and dynamic pressures encountered during the passes through the atmosphere.

3.6 TESTABILITY

3.6.1 Spacecraft Testability

Critical functions of the spacecraft and interactions between the spacecraft and the MOS, Delta II and DSN shall be testable at a level of assembly adequate to verify the function or interface to be tested. Normal and fault protection/correction hardware and software states and sequences shall be testable at the system level. The spacecraft and test system shall provide for safe ground testing of all hazardous spacecraft commands.

3.6.2 Government-Furnished Property Testability

The spacecraft design shall provide for functional verification of the payload operation at the system level. Hardware and software interactions among GFP subsystems and between GFP subsystems and the spacecraft shall be testable at the system level. Access to and electrical monitoring of payload interfaces shall be possible at the system level. Access for external stimulation of payload sensor elements shall be possible at the system level. Payload interfaces shall be designed so as not to limit installation/removal cycles or test duration at the system level.

3.7 DELTA COMPATIBILITY

The spacecraft, in its launch configuration, shall be fully compatible with the Delta launch vehicle.

3.8 LAUNCH SITE COMPATIBILITY

The spacecraft and GSE shall permit installation of a shroud or other equipment to maintain a cleanliness level of class 100,000 around the spacecraft during launch site processing.

The spacecraft and GSE shall be capable of routing and/or sending spacecraft telemetry and command data through a facility communication interface prior to encapsulation in the Delta payload fairing. After encapsulation in the payload fairing, all telemetry and command links shall be via a ground umbilical.

3.9 SAFETY

The spacecraft and GSE shall meet the requirements and be subject to the constraints set forth in ERR 127-1, "Range Safety," AFSPACECOM letter dated 23 November 1993, and GP1098, “KSC Ground Operations Safety Plan."

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SECTION 4

PAYLOAD DESCRIPTION AND ACCOMMODATION

4.1 GENERAL

The spacecraft design shall accommodate the payload as described in Table 4-1.

Table 4-1. Mars Global Surveyor Payload

Instrument	Abbreviation
Magnetometer/electron reflectometer	MAG/ER
Mars orbiter camera	MOC
Mars orbiter laser altimeter	MOLA
Mars relay	MR
Thermal emission spectrometer	TES
Ultrastable oscillator (radio science)	USO (RS)

4.2 MECHANICAL

The spacecraft shall accommodate the physical, orientation, and field of view requirements of the payload as defined in the payload Interface Control Documents (ICD) referenced in Section 2.

4.2.1 Configuration

The payload ICDs include a physical representation of each payload assembly. Three-view layouts with body-fixed orthogonal axes (X', Y', Z') are defined for each instrument assembly that has a specific orientation requirement. Physical dimensions are supplied for those assemblies that have no specific orientation requirements.

4.2.2 Orientation

All instruments for which body-fixed axes (X', Y', Z') are defined in the payload ICDs shall be oriented such that the +Z' axis is in the nadir direction, parallel to the orbital reference +Z axis. The +X' axis shall be in the downtrack direction, parallel to the orbital reference +X axis and velocity vector. The +Y' axis shall be normal to the +X' and +Z' axes, forming a right-handed coordinate system. Instruments shall be aligned relative to these axes to within 1 mrad with knowledge of the alignment to better than 0.5 mrad.

Note: In the payload ICDs, each instrument orientation is based on an orbit with the ascending orbit node on the dark side of the planet.

4.2.3 Instrument Location

Instrument locations on the spacecraft shall accommodate physical characteristics and fields-of-view as described in the payload ICDs.

4.2.4 Mass and Volume

The allocated mass and volume requirements for the instruments are specified in the payload ICDs. The spacecraft shall provide thermal blanketing and associated thermal hardware and equipment required for maintaining thermal control.

4.2.5 Fields of View

The view direction (VDIR), instrument field of view (IFOV), and stray radiation field of view (SRFOV) requirements are listed in the payload ICDs with respect to the instrument coordinates axes (X', Y', Z'). The spacecraft shall provide fields of view for the science instruments as specified.

4.2.6 Payload Sun Avoidance

To protect the instruments from damage, spacecraft operations shall insure that the +Z-axis is never pointed within 30 degrees of the sun, unless a slew rate greater than or equal to 0.4 degrees per second is maintained while the +Z-axis is within 30 degrees from the sun.

4.3 RADIO SCIENCE

The spacecraft telecommunications subsystem shall utilize the USO described in the USO ICD. The USO shall be located in a stable thermal environment ($\Delta T \leq 3°C/hr.$). The USO has two identical outputs (one for each transponder). When in the USO mode, the transponder will use the USO output as a frequency reference for the X-band downlink.

4.3.1 RS Atmospheric Observation Requirements

The spacecraft shall provide the capability for X-band transmission in the USO transponder mode from approximately 4 minutes prior to Earth occultation ingress to approximately 1 minute after, and from approximately 1 minute prior to Earth occultation egress to approximately 4 minutes after Earth occultation.[1] This capability shall be provided for each Earth occultation event during each tracking period of the mapping phase, within the limitations of the HGA gimbal travel. During this period the spacecraft shall provide the capability for X-band transmission in either the USO or two-way coherent transponder mode with telemetry modulation switched OFF and the transmitter ON.

The composite digital data stream and subcarrier shall be capable of being turned off during radio science atmospheric observations.

4.3.1.1 Frequency Stability of Downlink Carrier. When the downlink carrier is referenced to the USO for occultation measurements, the spacecraft shall not degrade the specified frequency stability and phase noise performance of the USO by more than 20% (1 dB). This shall include the effects of the telecommunications subsystem, motion of antenna relative to center of spacecraft mass, and the thermal/magnetic environment of the USO.

4.3.1.2 Spectral Purity. Modulation sidebands or spurious signals in the downlink signal transmitted by the spacecraft during occultation measurements shall be less than -60 dBc within 2000 to 500 Hz of the carrier and less than -70 dBc within 500 Hz of the carrier.

4.3.1.3 Amplitude Stability. The unmodeled contribution of spacecraft effects on received power of the downlink carrier shall be less than 0.1 dB over any 50 second interval during occultation measurement.

[1] This period of time is when the spacecraft to Earth line-of-sight passes between the surface and 200 km above the surface plus 100 seconds for RF calibration and timing pads to ensure the observation.

4.3.2 RS Gravity Field Measurements

4.3.2.1 Spacecraft Contribution to Doppler Error. When the spacecraft is seen at the DSN above 25° elevation (or approximately 50 dB/Hz), the spacecraft contribution shall be less than 0.1 mm/sec (3σ) unmodeled error in the Doppler tracking measurements with a 10 second integration time. This shall include the efects of telecommunication subsystem, instability of the onboard oscillator if coherent tracking is not available, and motion of the antenna relative to the spacecraft center of mass.

4.3.2.2 Spacecraft Telemetry. Spacecraft telemetry data for maneuvers, momentum unloading thruster firings, etc., shall be of sufficient accuracy to reconstruct spacecraft velocity changes to 1.0 mm/sec (3σ).

4.4 CHECKOUT AND CALIBRATION

4.4.1 Payload Cruise Engineering Checkout

The spacecraft shall provide for a payload engineering checkout to assess instrument health early in the cruise phase after TCM-1. All instruments shall be exercised during the checkout period. With the exception of the magnetometer and MOC calibrations, no specific maneuvers or spacecraft attitude are required.

4.4.2 Payload Calibration Requirements

The spacecraft shall provide for instrument health checks, calibrations, and a bake-out, as defined below, while maintaining adequate attitude control. Achieving the mapping orbit and maintenance of center-of-gravity control shall take precedence over these operations.

4.4.2.1 Magnetometer Calibrations. The MAG sensor calibration will require a minimum of 30 days of data collection (as continuously as possible) in the cruise phase and 30 days in the transition orbit, as permitted by aerobraking operations. Additionally, a calibration maneuver will be performed during cruise consisting of two consecutive revolutions around one spacecraft principal axis, followed by two revolutions around an axis nearly orthogonal to the first.

4.4.2.2 Mars Obsever Camera Calibrations. The MOC requires a period for bake-out and two calibrations during the cruise phase. The bake-out is required for a period of 60 days early in the mission. Prior to and subsequent to the bake-out period, two periods of up to seven days each of MOC calibration shall be provided to determine the effectiveness of the bake-out procedure. The two calibration periods each require rotation of the spacecraft and that instrument data be recorded and returned.

In addition, images of Mars shall be secured within two months of MOI and during the orbit insertion phase by properly orienting the spin plane of the spacecraft while preserving other instrument FOV requirements.

4.4.2.3 Calibration Data Return in Cruise. The spacecraft shall be capable of providing for low-rate real-time data return and playback of recorded instrument calibration data throughout the cruise phase as necessary to support check-out and calibration activites.

4.5 INSTRUMENT POWER

The spacecraft shall provide independently switchable and fused primary power lines for each instrument and as required to control replacement heaters, decontamination heaters, and pyrotechnic functions.

The spacecraft shall provide replacement heater power to each instrument during nonoperating conditions. The individual power requirements shall be specified in the payload ICDs.

4.5.1 Launch, Cruise, and Orbit Insertion Phase Power Requirements

The spacecraft shall be capable of providing power to the payload for necessary instrument bakeouts and calibrations as specified in Section 4.4. Cruise power requirements for instrument calibrations will not exceed the mapping power shown in Table 4-2, except for MOC bakeout. MOC bakeout will require additional power as detailed in the MOC payload ICD.

4.5.2 Mapping Phase Power

The spacecraft shall provide orbital average and peak power to each instrument as shown in Table 4-2. The total payload peak power capability shall be equal to the sum of the peak powers of all of the instruments flown. The peak power requirements in Table 4-2 do not apply to load current ripple, startup transients, or other transient loads.

4.5.3 Relay Phase Power

The spacecraft shall provide sufficient power to operate the MOC and MR as listed in Table 4-2.

4.5.4 Transient Characteristics

The spacecraft shall accommodate the instrument transient characteristics as specified in the payload ICDs.

Table 4-2. Payload Power Requirements

Instrument	Orbital Average Power (W)	Peak Power (W)	Operating Voltage (Vdc)
MAG/ER	4.63	4.63	28 ± 0.56
MOC and MR[a]	22.75	29.75	28 ± 6 and 28 ± 0.56
MOLA	34.20	35.94	28 ± 2
TES	12.29	18.20	26 - 32
USO	3.00	4.50[b]	24 - 30[c]

[a] The MOC average and peak allocations are 22.75 and 29.75 watts for the entire mapping phase of the mission with the MR off. When the MR is active, the MOC average and peak allocations drop to 10.25 and 17.25 watts, resulting in no net change for the combination of MOC and MR.

[b] The USO peak power is for a maximum of 2 hours at turn-on.

[c] The USO input voltage drift rate shall be less than one volt per hour.

4.6 INSTRUMENT THERMAL CONTROL

Spacecraft thermal control shall maintain the temperature application point of the USO, MR, TES, and MAG/ER within the ranges shown in Table 4-3. The temperature application point is defined as the location of the flight thermistor, as specified in the payload ICDs.

Table 4-3. Instrument Application Point Temperature Limits

	Temperature Limit (°C)	
Instrument	**Operating**	**Nonoperating**
USO	-20 to +30	-30 to +40
MR Electronics	-15 to +40	-30 to +50
MR Antenna Base	-100 to +100	-100 to +100
TES Electronics	-20 to +40	-20 to +50
MAG/ER Electronics	-10 to +40	-20 to +40
MAG Sensor	-20 to +45	-20 to +45
ER Sensor	-15 to +35	-30 to +35
MOC and MOLA	These instruments will maintain operating and nonoperating temperatures as stated in the instrument unique interface control documents.	

4.6.1 In-Flight Temperature Monitoring

The spacecraft shall monitor the temperature of each payload element. The location of these measurements shall be specified in the unique interface control document for each payload element. Such monitoring shall be incorporated into the spacecraft telemetry (ENG) so that the temperature information can be obtained, consistent with Paragraph 3.4.3.

4.7 ENVIRONMENTAL CONTROL

4.7.1 Magnetic and Electrostatic Fields

The MAG/ER experiment has three sensors: two magnetometer sensors and one electron reflectometer. Magnetic control requirements shall be established to ensure that the magnetic field due to any spacecraft-related cause is less than 3 nT (static) and less than 0.3 nT (peak-to-peak) for magnetic field variations at frequencies less than 10 Hz, as measured by the magnetometer most distant from the spacecraft center.

4.7.2 Payload Contamination Control

Contamination is defined as molecular and particulate material that has the potential to degrade the performance of the science instruments, attitude sensors, solar arrays, and thermal control surfaces. The spacecraft, together with test, handling, storage, launch and flight procedures, shall be designed to limit contamination on the sensitive instrument surfaces through the end of the mapping mission. The spacecraft external surfaces shall be visibly clean at payload encapsulation. Visibly clean is defined as the absence of all particulate and non-particulate matter visible to the normal unaided (except corrected vision) eye when inspected at a distance between 6 and 18 inches under a surface illumination of 100 ft-candles minimum. Particulate matter is defined as matter of miniature size with observable length, width and thickness. Non-particulate matter is a film without definite dimension.

4.7.2.1 <u>Nitrogen Purge Requirements.</u> The spacecraft design shall provide for continuous nitrogen purge for the TES, MOC and MOLA from instrument-spacecraft integration until Delta liftoff. The spacecraft contractor shall provide for nitrogen purge for the ER, with replenishment of the nitrogen environment around the ER every three days at a minimum.

4.7.3 Microphonics

The TES requires that the microphonic environment at the nadir panel instrument interface not exceed 0.005 g in the frequency range from 10 Hz to 120 Hz.

The MOC requires that the total magnitude of the acceleration at the instrument feet be below the values in Table 4-4 to keep the variation of pixel position from nominal to less than 2%.

Table 4-4. MOC Microphonics Limitations

Frequency Range	Acceleration Limit
2 Hz:	< 0.26 g;
2-20 Hz:	-25 dB/decade;
20-100 Hz:	<0.014 g;
100-400 Hz:	-30 dB/decade;
400-2000 Hz:	+80 dB/decade.

4.8 MARS RELAY REQUIREMENTS

The relay payload interface requirements are stated in the applicable MR and MOC ICDs.

SECTION 5

SPACECRAFT SUBSYSTEM FUNCTIONS

5.1 GENERAL

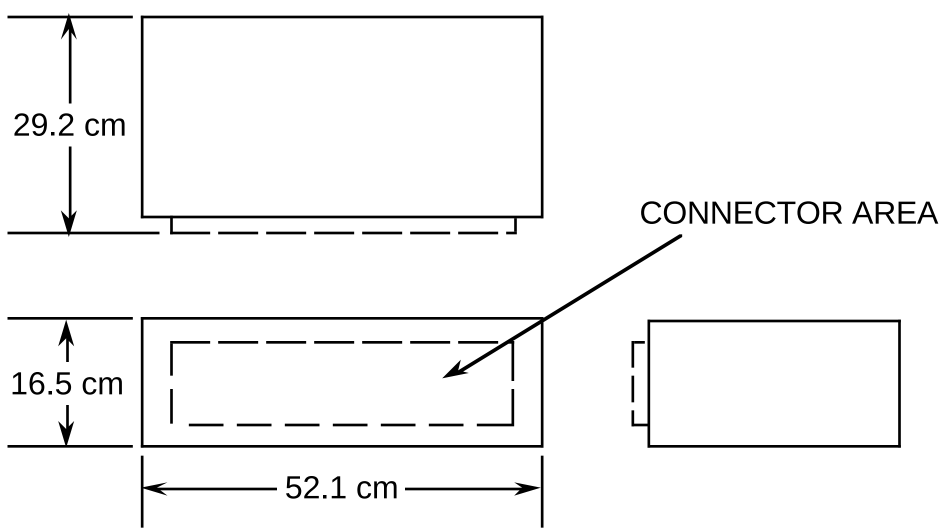
The spacecraft bus shall provide for the following subsystem functions:

(1) Command and data handling
(2) Telecommunications
(3) Attitude and articulation control
(4) Power
(5) Thermal control
(6) Structure, cabling, and mechanisms
(7) Propulsion and pyrotechnics

5.2 COMMAND AND DATA HANDLING

The spacecraft shall provide command and data handling (C&DH) for control and reconfiguration of the spacecraft and to permit science and engineering data collection.

The spacecraft shall utilize the GFP payload data subsystem (PDS), specified in JPL D-3419, Vol. 1, and JPL-D-6623. The spacecraft shall provide 5.25 W of power to the PDS at 28± 3 V, and up to 40 mA at 10 V. The spacecraft shall accommodate the PDS mechanical configuration given in Figure 5-1.



NOTES: SPACECRAFT CABLING LAYOUT SHOULD MINIMIZE TOTAL CABLE LENGTH BETWEEN PDS AND ALL INSTRUMENTS
MASS = 11.90 kg

Figure 5-1. Payload Data Subsystem Mechanical Configuration

5.2.1 Command Requirements

The spacecraft bus shall accept and process two classes of commands transmitted from the ground: real-time commands (RTCs) and stored-sequence command (SSC) files.

5.2.1.1 Real-Time Commands. RTCs result in an immediate action by the spacecraft bus as determined by the operation code within the command. RTCs may command spacecraft hardware or software activities, or may command PDS actions or payload activities via the PDS. RTCs may consist of multiple words, but shall not be executed until the spacecraft has validated receipt of the entire command string.

5.2.1.2 Stored Sequence Commands. An SSC file is a command file which is sent to the spacecraft during the course of the mission to effect the autonomous execution of a time-ordered series of spacecraft and/or instrument commands. An SSC file shall consist of a series of commands and/or calls to on-board stored scripts, each with an associated time tag. The time tags shall be either an absolute execution time or a time relative to the execution of the previous command or script call. Time tags shall be sized to encompass the duration of the mission.

A script is defined as a time-ordered series of spacecraft commands which is stored on-board in a library for use by other uplinked SSC files or by autonomous spacecraft housekeeping and fault protection functions.

The flight software shall provide the ability to specify a deterministic memory location for each SSC file or script at load time. There shall be an on onboard, automatic script load verification mechanism. It shall be possible to load a script or an SSC file and activate immediately upon satisfactory on-board verification, as well as by RTC or SSC file time-tagged activation commands.

Multiple scripts may execute simultaneously, and there shall be the capability to de-activate individual scripts.

The script execution software shall provide a capability to conditionally activate a script or terminate an already active script based on the following script flags:

1) RAM Safing Mode enabled flag
2) Late execution allowed flag
3) Active script restart allowed flag
4) Proceed on error flag
5) Load validation required flag

The flight software shall be capable of activating a previously uploaded script, specified by deterministic memory location, based on the following events:

1) Eclipse entry and exit detections
2) RAM Safing Mode Entry

5.2.1.3 Command Rate and Format. The spacecraft bus shall be capable of accepting in-flight-switchable uplink command rates as defined in DM 514438, "Deep Space Command Detector Unit National Aeronautics and Space Administration Design Requirement For." A low rate of 7.8125 b/s shall be used in an emergency or backup mode.

5.2.1.4 Command Storage and Timing. The C&DH shall be capable of storing spacecraft bus and at least 1500 PDS and science instrument commands, plus timing information for each command. The transfer of a single RTC to the PDS shall be completed within 2 seconds. The C&DH shall ensure that the transmissions of RTC and SSC contents to the PDS do not interrupt each other. The C&DH shall provide for a minimum of 5k words of memory in cruise and a minimum of 13k words of memory in mapping for storing command sequences and a script library.

5.2.1.5 Command Error Protection. The flight hardware and software shall be designed to preclude the unintentional execution of critical or irreversible commands. The spacecraft shall reject all invalid commands, whether they are RTCs or contained within SSCs.

5.2.1.6 Equator Crossing Command. During mapping operation, the spacecraft shall provide a time reference to the PDS containing both the upcoming and subsequent ascending node equator crossing times approximately 7 minutes before the first of these two events.

5.2.2 Data Handling Requirements

The spacecraft bus shall provide for the data modes and streams shown functionally in Figure 5-2. A description of these modes and data streams follows.

5.2.2.1 Data Streams. The spacecraft bus shall provide the following data streams:

(1) S&E-1. S&E-1 is a combined science and engineering stream for recording which is intended to permit the continuous collection of science observations. The combined data stream shall be sent to the C&DH by the PDS as a complete transfer frame that will conform to applicable CCSDS standards. The spacecraft shall provide packetized engineering data to the PDS for insertion into the S&E-1 data stream at a rate or equivalent rate not to exceed 256 b/s, and which shall meet the requirements of Paragraph 3.4.3.

(2) S&E-2. S&E-2 will be similar to S&E-1 except that it will utilize higher data rates and is intended for real-time transmission.

(3) ENG. ENG shall be an all-spacecraft bus engineering data stream; it will be assembled by the C&DH and shall conform to the requirements of applicable CCSDS standards. The C&DH shall provide for variable rates and telemetry content as required for all spacecraft operations. An engineering dwell mode shall be provided, wherein normal telemetry is suspended while the downlinked telemetry dwells on selected points until commanded otherwise.

5.2.2.2 Data Rates. Science instrument data rates input to the PDS are as defined in JPL D-4130, “PDS-Instruments Interface Requirements Document”. The PDS data rate outputs to the spacecraft are as defined in JPL D-3419, Vol. 1, “Payload Data Subsystem Functional Requirements Document.”

Data rates for the ENG data stream shall accommodate the emergency telemetry requirements specified in Paragraph 3.4.3.7 at 10 b/s, and shall provide for 256 b/s and 2 kb/s data rates for normal engineering-only telemetry.

5.2.2.3 Data Storage Capacity. The spacecraft bus shall provide the capability both to record and to playback the S&E-1 data stream. The capacity shall be adequate to record S&E-1 data at up to 16 ks/s data rate for a minimum of 24 hours. Simultaneous record and playback operations shall be as defined in JPL D-3419, Vol. 1, “PDS Functional Requirements Document.”

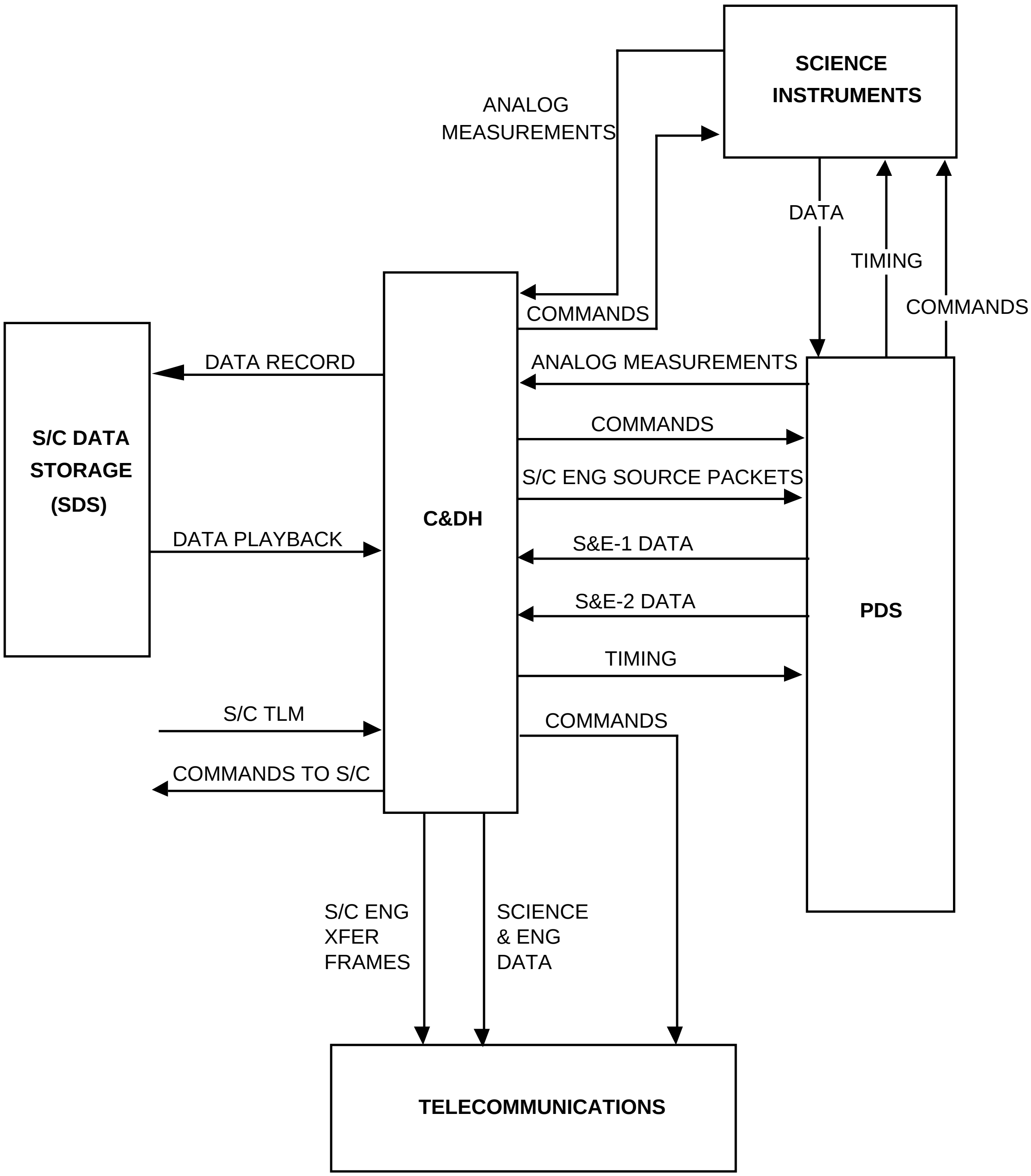


Figure 5-2. Command/Data Flow Diagram

5.2.2.4 <u>Output Data Channel.</u> The command and data handling subsystem shall produce one composite digital data stream to be routed to the telecommunications subsystem for transmission to Earth. S&E data shall be Reed-Solomon encoded. In addition, all data streams shall be convolutionally encoded and conform to the standards specified in the applicable CCSDS documentation.

5.2.2.5 <u>Spacecraft Clock.</u> The spacecraft bus shall provide the timing reference for the spacecraft. The spacecraft bus shall provide an unambiguous binary count to the PDS every 1 s and timing pulse every 125 ms. The time code supplied to the PDS and that which is embedded in the engineering source packets shall be coherent and derived from the same frequency reference. The time supplied shall be capable of being correlated with a known epoch to within 20 ms. The stability of the clock frequency source shall be such that the total drift over a 21-day period is predictable to an accuracy of 20 ms. The spacecraft shall provide an unambiguous count with each data frame in accordance with JPL CCSDS from launch through the end of the mission.

5.2.2.6 <u>Spacecraft Data Storage Management.</u> The spacecraft shall be capable of providing for on-board management of data storage and playback to prevent data losses resulting from Earth occultations and associated DSN lockup time requirements (approx. 5 min. required for lockup).

5.2.3 Flight Software and Fault Protection

5.2.3.1 <u>Flight Software Design</u>. The flight software shall provide for memory readout of any portion of memory of any specified size.

5.2.3.2 <u>Fault Protection Software.</u> In the event of a failure in a subsystem protected by fault protection software, the software shall provide sufficient data to reconstruct the autonomous fault protection response. All autonomous fault protection actions taken by flight software shall be reversible by ground command. Ground command override of any autonomous fault protection routine shall be possible.

5.3 TELECOMMUNICATIONS

The spacecraft shall provide for DSN-compatible X-band communications to and from the Earth for radiometric tracking, telemetry, commanding, and radio science as described in JPL document 810-5, "Deep Space Network/Flight Project Interface Design Handbook." The spacecraft shall also provide for a Ka-band link experiment. The telecommunications design shall be capable of simultaneous radiometric tracking, telemetry return, and commanding.

5.3.1 Receiving Requirements

At all times during the mission, at least one receiver/CDU shall be powered; it shall not be possible to turn both receivers off at the same time. If two receivers/CDUs are powered, the spacecraft shall provide the capability for flight software to select either receiver/CDU string for receipt of commands.

The spacecraft shall receive and phase lock to a modulated or unmodulated X-band uplink carrier signal. Channel frequency shall be in the range of 7145 to 7190 MHz. The spacecraft shall provide an automatic gain control (AGC) to adapt the receiver gain to the received X-band signal strength. The spacecraft receiver AGC and static phase error (SPE) shall be available in telemetry.

The spacecraft shall provide receiver phase lock operation to the uplink carrier, and the downlink carrier shall be generated from an internal auxiliary oscillator (AUX OSC) or from an external ultrastable oscillator (USO) during the two-way noncoherent tracking mode. The two-way noncoherent mode shall be selected by discrete spacecraft command.

The spacecraft shall provide the capability for “turnaround” ranging. In this mode, the uplink ranging data shall be demodulated from the X-band uplink carrier and remodulated on the X-band downlink carrier. The spacecraft shall provide a turnaround ranging mode on/off capability.

Receiver tracking threshold shall be ≤154.7 dBm. Acquisition and tracking rate with an unmodulated signal level greater than -90 dBm shall be 550 Hz/s minimum.

5.3.1.1 Cruise, Orbit Insertion, Mapping and Relay Phases. The spacecraft shall be capable of receiving an X-band uplink carrier phase-modulated with a command subcarrier and ranging data from the DSN. The spacecraft shall be capable of providing a primary uplink path with a minimum G/T of -17.5 dB/K referenced to the antenna output terminals. This minimum value is sufficient for simultaneous high-rate commanding and two-way Doppler tracking via a 34 meter Deep Space Station (DSS) at maximum range.

5.3.1.2 Emergency Receiving Capability. The spacecraft shall provide a reduced-performance uplink with a minimum G/T (mean-3σ) of -28.2 dB/K referenced to the low-gain antenna input terminals. This minimum value is sufficient for emergency low-rate commanding via the 34 meter DSS at maximum range.

5.3.2 Transmitting Requirements

The spacecraft shall utilize the received uplink to coherently generate the X-band downlink carrier signal at a transmit/receive frequency ratio of 880/749 (two-way tracking). Transfer to AUX OSC or USO operation shall be automatic upon loss of receiver phase lock. Channel frequency shall be in the range of 8400 to 8450 MHz.

The spacecraft shall generate a downlink carrier from either an AUX OSC or from a USO, when the receiver is not phase-locked to an X-band uplink signal (one-way tracking). Operation in the USO mode shall be by discrete spacecraft command. The spacecraft shall also provide an exciter ON/OFF function.

5.3.2.1 DSN Initial Acquisition. During the period from Delta II third stage separation and attitude initialization until DSN initial acquisition has been verified, the spacecraft shall provide a minimum effective isotropic radiated power (EIRP) of 37.5 dBm. DSN acquisition will be within 30 minutes after spacecraft comes in view of DSN station with spacecraft transmitter on.

5.3.2.2 Cruise and Orbit Insertion Transmitting Requirements. During cruise and orbit insertion phases the spacecraft shall support a 4 kb/s downlink, and shall provide a minimum EIRP as indicated in Figure 5-3.

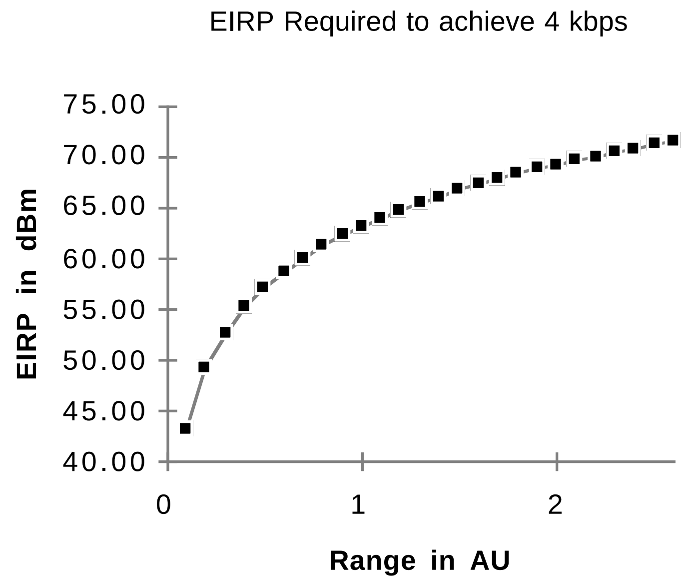


Figure 5.3. Cruise Minimum EIRP Requirements

5.3.2.3 Mapping Transmitting Requirements. From the beginning of the mapping phase through the end of the mission, the spacecraft shall be capable of providing an EIRP that is sufficient to meet navigation requirements defined in Section 3.3 and engineering and science telemetry data return requirements defined in Section 3.4. The required EIRP at maximum range shall be derived from the data in Figure 5-4 on the basis of minimum S&E data rates described in JPL D-3419, Vol. 1, "Payload Data Subsystem Functional Requirements Document."

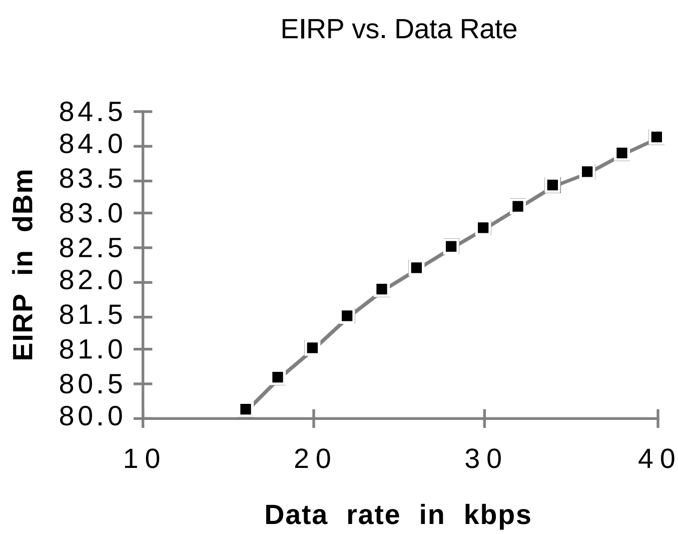


Figure 5-4. Minimum EIRP vs. Data Rate

5.3.2.4 Emergency Telemetry Capability. For emergency engineering telemetry, the spacecraft shall provide a minimum EIRP of 46.0 dBm at the emergency rate.

5.3.2.5 Telemetry Modulation. The spacecraft shall provide the capability to phase modulate the RF drive for the X-band downlink carrier with composite telemetry supplied by the spacecraft telemetry data system.

The spacecraft shall provide two square-wave subcarriers: one for symbol rates greater than 500 symbols per second and one for symbol rates of 500 symbols per second or less. The subcarriers shall be biphase modulated with convolutionally encoded data. For each subcarrier, there shall be a range of modulation indices which are in-flight selectable independent of the data rate. Channel coding shall conform to the standards specified in the CCSDS standards. The subcarrier frequencies shall be greater than 20 kHz and shall be at least 1.5 times the maximum downlink symbol rate in symbols per second. For symbol rates of 500 symbols per second or less, the subcarrier frequency shall not exceed 40 kHz.

5.3.2.6 Ka-Band Link Experiment. The spacecraft shall accommodate a Ka-band link experiment (KABLE) that will generate a coherent downlink carrier at 32 GHz ± 0.25 GHz with minimum impact on the performance and reliability of the X-band communication link. The KABLE spaceborne hardware shall not be considered mission critical.

5.4 ATTITUDE AND ARTICULATION CONTROL SUBSYSTEM (AACS)

The spacecraft bus shall have sufficient control authority to automatically maintain attitude orientation and stability of the spacecraft during all phases of the mission, including and following separation from the Delta II, and to support the required spacecraft functions associated with communications, power, thermal control, and propulsive maneuvers.

During the mapping phase, the nadir orientation of the spacecraft shall be maintained as specified below. The AACS shall provide a spacecraft attitude that supports continuous and simultaneous data taking by all science instruments during mapping.

The AACS shall also be capable of pointing the body mounted science instruments at arbitrary targets in the celestial frame. The pointing profile shall be able to incorporate an open-loop slew.

5.4.1 Orbital Reference Coordinate System

The orbital reference coordinate system in Mars orbit is shown in Figure 5-5.

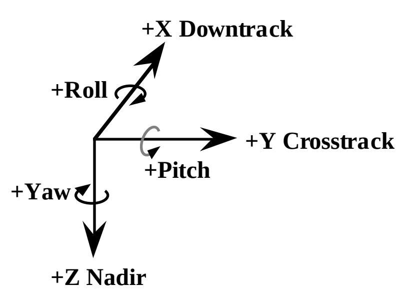


Figure 5-5: Orbital Reference Coordinate System

The fundamental planet reference direction is the nadir direction. The +Z axis (nadir direction) is defined by a line passing through the spacecraft perpendicular to the Mars mapping reference spheroid (polar radius = 3375.7 km, equatorial radius = 3393.4 km). The +X axis is defined to lie in a plane perpendicular to the nadir direction and along the projection of the velocity vector on this plane. The +Y axis also lies in this plane and is orthogonal to both the +Z and +X axes, forming a right-handed coordinate system. As defined, the +X axis will be close to, but not always coincident with, the direction of the spacecraft velocity vector; and the +Y axis will be close to, but not always coincident with, the orbit normal.

"Yaw" shall be taken to refer to a rotation about the Z axis, "roll" to a rotation about the X axis, and "pitch" to a rotation about the Y axis. Pointing control (in terms of roll, pitch, and yaw) shall refer to the +X, +Y, and +Z axes as defined above.

5.4.2 Pointing Control, Knowledge, and Stability

5.4.2.1 Instrument Pointing Control. During mapping, the spacecraft shall be able to control the pointing of the baseplate of the body-mounted science instruments to within ±10 mrad (per axis, 3σ) with respect to the orbital reference coordinate system.

5.4.2.2 Instrument Inertial Pointing Control. When pointing at arbitrary celestial targets, the spacecraft bus should, as a goal, be able to control the pointing of the baseplate of body mounted science instruments to within ±6 mrad (per axis, 3σ) with respect to the celestial frame. The AACS should be capable of holding an arbitrary spacecraft attitude for at least 1 hour, barring orientations that violate sun avoidance constraints.

5.4.2.3 Instrument Pointing Knowledge. Pointing knowledge error is defined as the difference between the actual pointing vector and the estimated pointing vector. The spacecraft bus shall provide sufficient engineering telemetry to obtain a non-real-time, reconstructed pointing knowledge error of the body-mounted instruments during mapping of less than 3 mrad (per axis, 3σ). The spacecraft bus shall provide engineering telemetry to obtain a non-real-time, reconstructed pointing knowledge of the solar array-mounted magnetometer. Pointing knowledge shall be defined relative to the orbital reference coordinate system.

5.4.2.4 Instrument Pointing Stability. During the mapping phase of the mission, there are two spacecraft bus Areodetic pointing stability requirements: one for short instrument integration times (0.5 s), and one for longer integration times (12 s). Over a 0.5 second period of time, the attitude excursions shall be less than 0.5 mrad in pitch and roll respectively, and 1.0 mrad in yaw. Over a 12 second period of time, the attitude excursions shall be less than 3.0 mrad in pitch, roll, and yaw respectively.

Attitude angular rates outside these values shall be minimized in number and duration. Additionally, the attitude angular rate information shall be derivable from the engineering telemetry.

5.4.2.5 Antenna Pointing Control, Knowledge and Stability. During periods when antenna pointing is required, the spacecraft bus shall be capable of automatically controlling the pointing of its antenna such that the transmitting and receiving requirements of Sections 3.3 and 3.4 are met.

5.5 POWER

5.5.1 Capability and Distribution

The spacecraft shall be capable of generating, storing, supplying, controlling, converting, regulating, and distributing all primary electrical power required for spacecraft functions. Primary electrical power consists of all power from any power source(s) to the primary side of any DC-DC converters. This capability is required continuously from spacecraft internal power-on prior to launch through all subsequent mission phases. The power subsystem shall support normal spacecraft operations in the worst-case orbit conditions.

5.5.2 Power Requirements

5.5.2.1 Launch, Cruise, and Orbit Insertion Phase Power Requirements. The spacecraft shall provide the necessary power for spacecraft operation, maneuvers, and instrument activities.

5.5.2.2 Mapping and Relay Phase Power Requirements. The spacecraft shall be able to provide continuous power to the payload, on an orbital average basis, as specified in Section 4. Power distributed to the payload shall meet the payload power regulation requirements in Section 4.

5.5.3 External Power

The spacecraft shall be capable of accepting externally provided power during ground and launch operations as necessary. Primary power shall be provided to the spacecraft by GSE during ground testing. This power shall mimic each of the power subsystem sources' output characteristics over their full operating ranges. The transfer from GSE-supplied external power to spacecraft internal power shall be accomplished without power interruption and shall be reversible and repeatable.

The spacecraft shall incorporate circuitry to protect it from short-circuit-type faults in the externally provided power source.

5.5.4 Overload Protection

Appropriate overload protection shall be provided by the spacecraft for all electrical loads external to the power subsystem.

5.5.5 Grounding

No single fault from high-side or low-side of the primary power distribution system to the chassis shall cause a mission-critical failure. The primary power distribution system consists of all hardware and cabling required to distribute electrical power to the spacecraft loads, from the power source(s) up to and including the primary side of any DC-DC converters.

5.6 THERMAL CONTROL

5.6.1 Capabilities

The capability to maintain spacecraft components within their flight-allowable operating and/or nonoperating temperature ranges shall be provided for all mission phases.

Thermal control surfaces shall not have a direct line-of-sight to the exit aperture of any propulsion thruster.

The spacecraft shall provide for instrument thermal control. The thermal interface requirements between the instruments and the spacecraft are defined in the unique interface control documents for each instrument. The spacecraft shall maintain the operating and nonoperating temperature of each instrument within allowable temperature levels as defined in Section 4.6.

5.6.2 Distortions

Attitude control pointing error budgets for the spacecraft shall include misalignments resulting from thermal distortions.

5.7 STRUCTURE, CABLING, AND MECHANISMS

5.7.1 General Design Criteria

The spacecraft structure shall perform without compromising the mission or the functional integrity of the spacecraft during all phases of the mission.

The minimum factor of safety for ultimate stress (FS_u) on the structure shall be 1.25. Unstressed structural members may require higher factor of safety. Analytical flight system safety factor requirements are contained in Table 5-1 of JPL D-11510 , “Spacecraft Performance Assurance Provisions.”

5.7.2 Mass

The spacecraft bus shall be designed to accommodate 75.6 kg of payload mass, plus an initial contingency of 2%.

5.7.3 Alignments

Attitude control pointing error budgets for the spacecraft shall include any structural misalignments.

5.8 PROPULSION AND PYROTECHNICS

5.8.1 Propulsion

The spacecraft shall provide the propulsive capability to (1) maintain spacecraft attitude control throughout all phases of the mission, (2) execute TCMs, (3) place the spacecraft into the capture orbit at Mars, (4) support insertion into the mapping orbit, (5) maintain the proper orbit for the mission duration, and (6) place the spacecraft in the planetary quarantine orbit at the end of the mission, if necessary.

Liquid propulsion subsystems shall include protection against overpressurization during the long periods between propulsive maneuvers. Propellant vapor and liquid shall be isolated from the pressurization system prior to initial system pressurization. The propulsion system shall protect against fuel or oxidizer migration and condensation in pressurization lines for all mission phases.

5.8.2 Pyrotechnics

Pyrotechnics systems, excluding initiators, shall meet the design requirements portion of MIL-STD-1576, “Electroexplosive Subsystem Safety Requirements and Test Methods for Space Subsystems.”